EXHIBIT III

MARS SCIENCE LABORATORY
TERMINAL DESCENT SENSOR
SPECIFICATION

Dated: May 31, 2005

Exhibit III

Mars Science Laboratory Terminal Descent Sensor Task Specification

1 SCOPE

This Specification establishes the requirements for design, fabrication, test, product assurance, performance, and environment of a Ku-band Doppler velocimeter/altimeter for guiding the descent and landing of the rover onboard the Mars Science Laboratory spacecraft.

2 APPLICABLE DOCUMENTS

The following documents of the issue specified form a part of this specification. In case of conflict between the requirements of this document and the requirements of any documents referenced herein, the conflict shall be referred to the JPL Contract Technical Manager for resolution.

JPL Specifications	Rev.	Title
JPL D-560	TBD	JPL Standard for System Safety
JPL D-5703	TBD	JPL Reliability Analysis Handbook
JPL D-8545	D	JPL Electronic Parts Derating Guidelines
JPL D-21382	TBD	Mars Science Laboratory Environmental Requirements Document (ERD)
JPL D-27175	TBD	Mars Science Laboratory Mission Assurance Plan (MAP)
JPL D-19426	TBD	Plastic Encapsulated Microcircuits (PEMs) Reliability/Usage Guidelines for Space Applications
JPL STD 00009	B	JPL Engineering Standard

3 BACKGROUND / MISSION(S) INFORMATION

Information on future NASA Office of Space Science (OSS) Mars Exploration Missions can be found at the URL: http://marsprogram.jpl.nasa.gov/missions/

The overall MSL science objective is to explore and quantitatively assess a local region(s) on the Martian surface as a potential habitat for life, past or present. This mission will use a variety of instruments carried on a rover platform that is expected to remain active for one Mars year.

MSL plans to place a scientific rover on Mars using a soft-landing approach similar to that used by the Viking missions. The spacecraft is planned for launch in the 2009 opportunity with the arrival at Mars occurring sometime in 2010. In order to accomplish this, the Entry, Descent, and Landing (EDL) system must be able to accurately measure altitude and 3-axis velocity (i.e., horizontal and vertical velocity) beginning at several kilometers and continuing all the way down to a few meters above the surface.

3.1 GENERAL FUNCTIONAL DESCRIPTION

Figure 1 shows the nominal timeline of the landing vehicle during the EDL phase. Upon heatshield jettison, the terminal descent sensor (TDS) is activated and the mobility system (the rover wheels and suspension system) is deployed. The TDS will measure 3-axis velocity and altitude from about 100 m/s and 4 km altitude to rover touchdown. The supersonic parachute and backshell are jettisoned before the start of powered descent. During powered descent, six throttle-able engines on the descent stage of the spacecraft are used to null out the horizontal velocity components and decelerate the spacecraft to about 3 m/s in vertical velocity about 35 m above the surface of Mars. At this point, the sky crane maneuver begins by lowering the rover down from the descent stage on 7.5 m long tethers. Figure 2 shows the spacecraft with the rover attached to the descent stage via a triple bridle. When the descent stage senses rover touchdown, the bridles are cut and the descent stage flies away past some minimum safe distance.

The TDS will be mounted at one of the following locations:

1) The bottom of the rover.
2) Somewhere on the descent stage.

The size of the TDS and the cone of clearance for the radar signal are crucial in determining the ultimate location of the sensor, but the preferred location is somewhere on the descent stage.

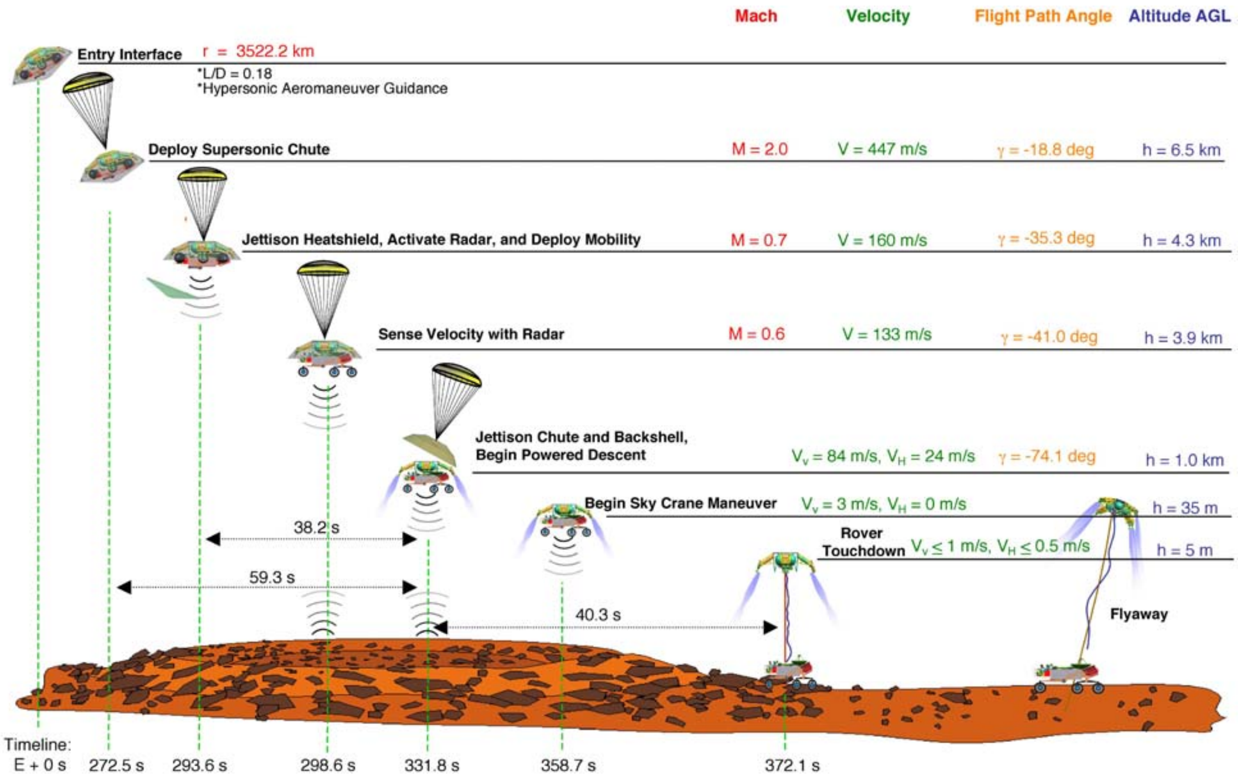

Figure 1. Nominal timeline of the spacecraft during the EDL phase.

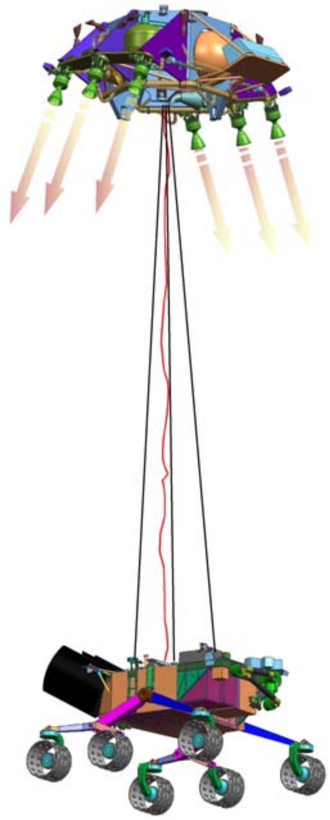

Figure 2. The rover is attached to the descent stage via a triple bridle

The TDS is expected to output per beam ground relative velocity and slant range measurements and their respective validity flags. The spacecraft computer has a Kalman filter that combines the TDS and IMU (Inertial Measurement Unit) measurements and estimates spacecraft 3-axis velocity and altitude in a local coordinate frame. To keep the EDL system simple, no IMU or estimated velocity data will be fed to the TDS.

While in this RFP the MSL project requires the TDS to output per-beam velocities (computation of 3-axis velocity is done by the spacecraft computer), the "Instantaneous Velocity Measurement Noise" requirement" (4.2.1) is stated as a 3-axis velocity measurement in the sensor frame. The reason for this apparent inconsistency is that the conversion of the 3-axis velocity requirement into per-beam requirement is dependent on the beam geometry, which the vendor is free to select. Therefore, the vendors are expected to combine the per beam measurements into 3-axis velocity components just for the purpose of determining if the "Instantaneous Velocity Measurement Noise" requirement is met. It is also important to note that MSL is levying an instantaneous measurement noise requirement, instead of the 10 nautical miles average that is commonly used in the helicopter radar industry, because the Kalman filter that implements the TDS-IMU sensor function is the responsibility of the MSL project.

3.2 Requirements Verification

The instrument, environmental, and mission assurance requirements are detailed in sections 4, 5, and 6. A preliminary requirements verification table is shown in Appendix B to demonstrate the verification approach that is required of each TDS requirement.

4 Instrument Requirements

The instrument requirements are subdivided into functional, performance, operational envelope, interface, physical, and life expectancy requirements.

4.1 FUNCTIONAL REQUIREMENTS

4.1.1 System Architecture

The TDS shall be a single box Ku-band radar.

4.1.2 Beam Configuration

The TDS shall have four beams arranged in a Janus configuration.

4.1.3 Beam Width

Each TDS beam shall have a one-way 3-dB beamwidth of 5º or less in both antenna planes.

4.1.4 Antenna Architecture

The TDS antenna(s) shall be integrated to the electronics box.

4.1.5 Velocimetry

The TDS shall measure, for each beam, spacecraft ground-relative velocity along the beam direction.

4.1.6 Altimetry

The TDS shall measure, for each beam, slant range to the ground.

4.1.7 Measurement Output

The TDS shall output a single beam measurement that is either velocity or slant range, where the beam and measurement type are selected by a spacecraft-commanded duty cycle.

4.1.8 Independent Operation

The TDS shall meet its functional and performance requirements without external IMU information.

4.1.9 Software Interface

The TDS shall have the capability to reload its software and parameters via the TDS-to-spacecraft data interface.

4.2 PERFORMANCE REQUIREMENTS

The performance requirements for the TDS are:

4.2.1 Instantaneous Velocity Measurement Noise

The TDS shall have an **instantaneous** 3-axis velocity measurement error in the sensor frame that is less than or equal to the following (all 3-sigma):

4.2.1.1 V_x

2.0% of $V_{total} + 0.50$ m/s

4.2.1.2 V_y

2.0% of $V_{total} + 0.50$ m/s

4.2.1.3 V_z

1.0% of $V_{total} + 0.20$ m/s

where $V_{total} = \mathrm{sqrt}(V_x^2 + V_y^2 + V_z^2)$.

Note: It is assumed here that the instantaneous velocity measurements are statistically uncorrelated from measurement to measurement, zero mean, and produced at a rate of at least 5 Hz.

4.2.2 Bias Speed Offset

The TDS shall have a bias speed offset that is less than or equal to 0.03 m/s (3-sigma) per axis measured over a 60-second period.

4.2.3 Bias Scale Factor

The TDS shall have a bias scale factor that is less than or equal to 0.25% (3-sigma) per axis assuming no terrain effects.

4.2.4 Quantization Accuracy

4.2.4.1 Velocity Quantization

The TDS shall have a beam velocity measurement quantization error that is less than or equal to 0.01 m/s.

4.2.4.2 Altimetry Quantization

The TDS shall have a slant range measurement quantization error that is less than or equal to 0.25 m.

4.2.5 Instantaneous Slant Range Measurement

For each beam, the TDS shall have a slant range measurement error (including statistical and deterministic error sources) that is less than or equal to 5% of actual slant range + 1 m (3-sigma).

4.2.6 Beam Pointing Knowledge

The electrical boresight of each beam in the TDS reference frame shall be known to within 2.5 mrad (3-sigma).

4.2.7 Acquisition Time

The TDS shall acquire lock in less than 1 second from reception of TDS Turn-On command.

4.2.8 Output Rate

The TDS shall produce a per-beam slant range and/or velocity measurement every 50 ms or less.

Note: In velocity-only mode, this implies that a 3-axis velocity measurement can be constructed based on 4 new *per beam* velocity measurements every 200 ms.

4.2.9 Time Tag Knowledge Error

The TDS time tag shall be known to less than or equal to 10 milliseconds.

4.2.10 Measurement Latency

The TDS shall make the last measurement available in less than or equal to 0.5 seconds after the measurement is taken.

4.3 OPERATIONAL ENVELOPE

The TDS shall meet the performance requirements in section 4.2 in the following operational envelope:

4.3.1 Altitude Range

3500 m to 10 m

4.3.2 Vertical Velocity Range

-20 to +120 m/s (+Z-axis is downward)

4.3.3 Horizontal Velocity Range

±50 m/s

4.3.4 Attitude Range

Off-nadir angles (angle between the TDS Z-axis and spacecraft nadir):

4.3.4.1 For 3000 m to 2000 m altitude

Up to ±35º

4.3.4.2 For 2000 m to 1000 m altitude

Up to ±40º

4.3.4.3 For 1000 m to 10 m

Up to ±50º

4.3.5 Attitude Rate Limits

Up to ±50º/sec

4.3.6 Linear Acceleration Limits

4.3.6.1 Vertical Acceleration

Up to ±30 m/s^2 in the vertical direction

4.3.6.2 Horizontal Acceleration

Up to ±10 m/s^2 in the horizontal direction

4.3.7 Ground Slopes

4.3.7.1 Long Wavelength

Up to 10º on a 2 km length scale

4.3.7.2 Short Wavelength

Up to 30º on a 20 m length scale

4.3.8 Unambiguous Altitude

If the TDS generates a valid slant range measurement (i.e. validity flag is positive) when the spacecraft is at an altitude that exceeds the maximum required altitude (see 4.3.1), then the measurement shall be unambiguous (but may not have to meet performance specification) as long as the altitude does not exceed 10 km. This requirement is necessary in the event that the heatshield is jettisoned at a higher altitude than anticipated. Note: the TDS is not required to produce a **valid** measurement at altitude exceeding 3500 m.

4.3.9 Unambiguous Velocity

If the TDS generates a valid beam velocity measurement (i.e. validity flag is positive) when the spacecraft is traveling at a velocity that exceeds the maximum required velocity (see 4.3.2 and 4.3.3), the measurement shall be unambiguous as long as the velocity does not exceed 300 m/s. Note: the velocity measurement may not need to be accurate, just as long as it is not wrapped.

4.4 INTERFACE REQUIREMENTS

4.4.1 Velocimetry

The TDS per beam velocity measurement shall consist of the following:

4.4.1.1 Time Tag

The time corresponding to the measurement

4.4.1.2 Validity Flag

The flag indicating that the measurement has passed TDS single beam internal validity checks

4.4.1.3 Beam Number

The beam associated with the measurement

4.4.1.4 Velocity Measurement

A signed floating point number in units of m/s

4.4.2 Altimetry

The TDS per beam slant range measurement shall consist of the following:

4.4.2.1 Time Tag

The time corresponding to the measurement

4.4.2.2 Validity Flag

The flag indicating that the measurement has passed TDS single beam internal validity checks

4.4.2.3 Beam Number

The beam associated with the measurement

4.4.2.4 Slant Range Measurement

A positive floating point number in units of meters

4.4.3 Data Interface Standard

The TDS shall interface with the spacecraft data bus via dual redundant MIL-STD-1553B.

4.4.4 Bus Voltage

The TDS shall interface with the spacecraft power bus via 28 VDC +/- 7 V.

4.5 PHYSICAL REQUIREMENTS

4.5.1 Dimensions

4.5.1.1 Footprint

The footprint of the TDS shall not exceed 40 cm x 40 cm.

4.5.1.2 Height

The height of the TDS shall not exceed 6 cm.

4.5.2 Mass

The mass of the TDS shall not exceed 10 kg.

4.5.3 Power

The average power consumption of the TDS shall not exceed 100 W.

4.6 LIFE EXPECTANCY

4.6.1 Operating Life

The operating life (Mean Time Between Failures) of the TDS shall be, at the minimum, 3000 hours in the environment specified in Section 5 of this document.

4.6.2 Thermal Cycle Life

The operational thermal cycle life of the TDS shall be a minimum of 8 cycles over the temperature range specified in 5.2.1.2 for the Flight Unit.

4.6.3 Shelf Life

The shelf life of the TDS shall be a minimum of 5 years when packaged per the packaging requirements in the Statement of Work.

4.7 MECHANICAL REQUIREMENTS

4.7.1 Alignment Markings

The TDS shall have a corner cube, fiducial marks, or alignment pins that define the TDS reference frame.

4.7.2 Identification and Marking Methods

The TDS assembly shall be marked with an assembly part number and a serial number.

4.7.3 Connectors

All TDS connectors shall be placed to facilitate ease of interconnection and subassembly replacement. The location of the connectors shall be shown on the ICD. The connectors used on the TDS shall be in accordance to the JPL connector specification listed in section 6.4 of JPL Engineering Standard (JPL STD00009 Rev. B).

4.7.4 Connector Retention

The connectors shall not use the body holes with fasteners for retention.

4.7.5 Venting

The TDS shall be vented to prevent damage during the pressure transitions during launch and Mars descent where the respective pressure decay rates are specified in 4.7.2 and 4.7.3 of JPL D-21382, the MSL Project Environmental Requirements Document.

4.7.6 Maintainability

The TDS shall be designed such that scheduled maintenance and repair, or adjustments are not required.

4.7.7 Workmanship

Workmanship relating to the application of standard processes used in the fabrication of the TDS shall conform to the requirements of the process specifications called out on the assembly drawing. Critical steps of fabrication that are item specific shall be detailed in drawing notes that shall include appropriate workmanship criteria.

4.7.8 Contamination Control

The TDS shall meet contamination control requirements specified in Section 6 of JPL D-27175 (MAP).

4.8 ELECTRICAL REQUIREMENTS

The TDS shall verify the electromagnetic compliance (EMC) requirements listed in Section 4.13.3 of JPL D-21382 (ERD). These requirements include:

4.8.1 Conducted Emissions

4.8.2 Radiated Emissions

4.8.3 Radiated Susceptibility

4.8.4 Conducted Susceptibility

4.8.5 Rejection of Undesired Signals

4.8.6 Power Line Ramp Voltage

4.8.7 Bonding, Isolation, and Grounding

4.8.8 Shielding

4.8.9 Cabling

Radiated emissions and susceptibility test shall be performed with flight-like test cables including representative wire shields, circuit references, and shield termination methods.

4.9 SAFETY

4.9.1 Design Principles

The design of the TDS shall be free from conditions that could cause injury to personnel or damage to the TDS or interfacing equipment

4.9.2 System Standards

The TDS subcontractor shall meet the Safety requirements of JPL D-560, JPL Standard for System Safety.

5 ENVIRONMENTS

The TDS shall be designed to meet the performance and operational envelope requirements (Sections 4.2 and 4.3) when subjected to the following environments in any order of sequence or combination thereof.

5.1 MARS ENVIRONMENTS

The TDS shall be capable of continuous operation, without damage, from a vacuum state to Mars atmosphere during the Mars Entry, Descent, and Landing (EDL) phase. The EDL environments are:

5.1.1 Operational Flight Temperature

Operational flight temperature on Mars: -40ºC to +50ºC

5.1.2 Barometric Pressure

Barometric Pressure on Mars: 2-12 torr

5.1.3 Relative Humidity

Relative humidity on Mars: 0 percent

5.1.4 Atmospheric Composition

Atmospheric composition on Mars: 95.5% CO_2, 2.7% N_2, 1.6% Ar, 0.2% Trace

5.1.5 Radar Backscatter Characteristics

Predictions of Ku-band backscatter cross-section were made based on Earth-based radar observations of Mars with X-band circularly polarized beams. Figure 3 shows the range of expected X-band backscatter behaviors for Mars in dB. The “worst case scenario” for radar scattering from Mars is represented by the “Stealth” region, a continent-sized area with less than 1% backscatter cross-section as mapped with the Goldstone/Very Large Array bistatic radar experiment. Globally, the Ku-band specific backscatter cross-section is expected to be greater than that for the smooth same-sense circular polarization (SC) behavior at Meridiani (red line in Figure 3).

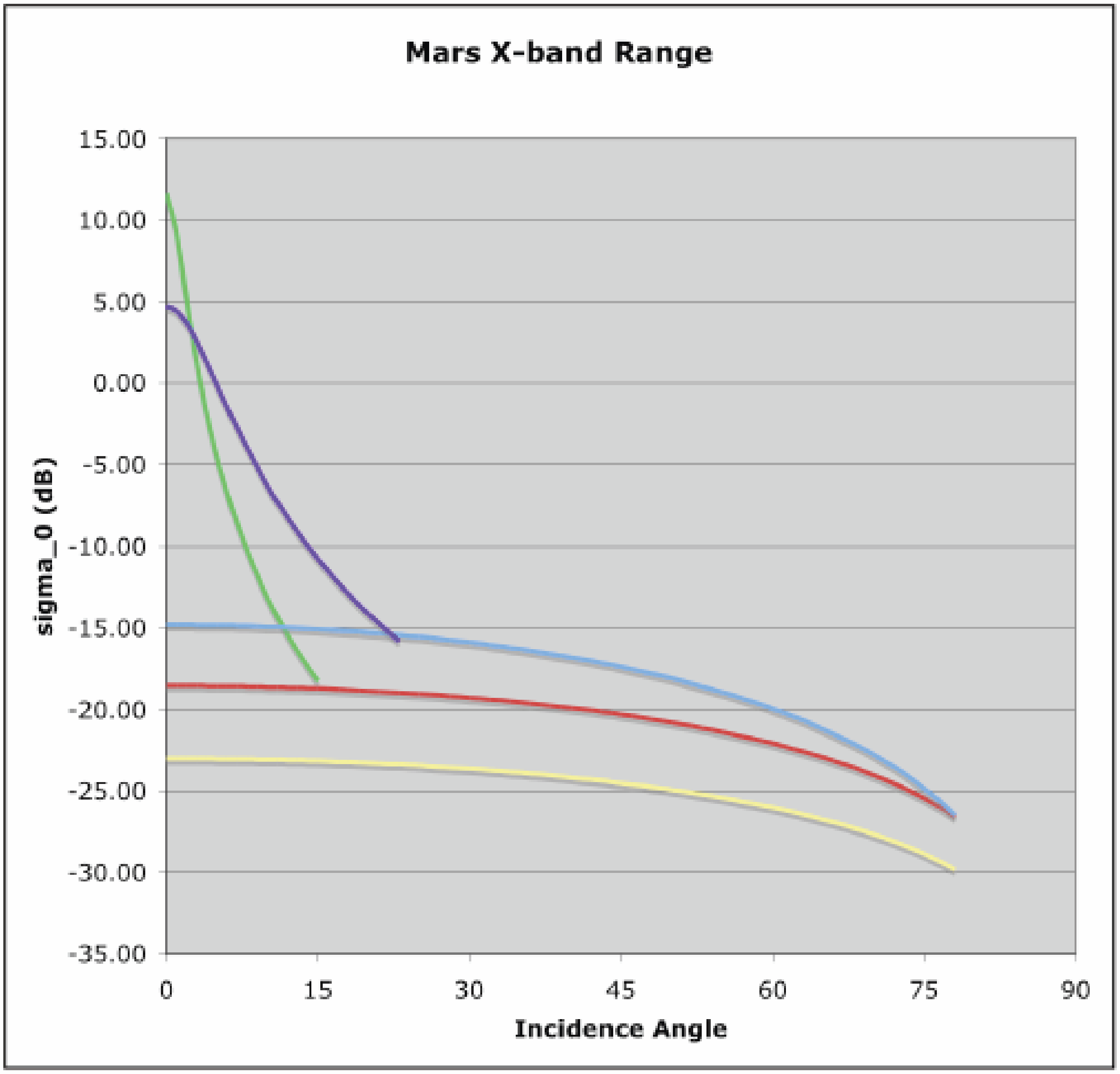

Figure 3. Range of X-band backscatter behaviors for Mars in dB. Green: smooth regions' quasi-specular behavior (Meridiani Planum). Purple: rougher regions' quasi-specular behavior (Hesperian channels). Blue: Mars average SC backscatter behavior. Red: SC behavior of the smooth Meridiani plains. Yellow: estimate of "stealth" behavior, 3 dB down from SNR limit.

5.2 ENVIRONMENTAL DESIGN REQUIREMENTS

The TDS shall be designed to operate over all combinations of the environmental requirements specified in JPL D-21382, the MSL Project Environmental Requirements Document (ERD). The following is a summary of these requirements:

5.2.1 Operational Temperature Limits

The temperatures given below, except where otherwise indicated, refer to the temperature of an isothermal base plate on which the TDS is to be mounted.

5.2.1.1 Allowable Flight Temperature

-40ºC to +50ºC

5.2.1.2 Flight Acceptance (FA)

-45ºC to +55ºC

5.2.1.3 Qualification (Qual)

-55ºC to +70ºC

5.2.2 Thermal Vacuum

The TDS performance according to specification shall be verified in a vacuum over the temperature ranges specified in paragraph 5.2.1. The thermal vacuum environments are listed in Table 2.2 of JPL D-21382 (ERD). The TDS shall be tested to the temperature profile shown in Figure 6 for qualification testing and flight acceptance testing.

5.2.3 Random Vibration

The TDS shall not be damaged or reduced in performance in any way after exposure to the random vibration environment described below. These test levels are applied at the mounting points of the TDS in each of the 3 mutually orthogonal axes with the TDS powered off for a duration of 2 minutes per axis for the qualification unit and 1 minute per axis for the flight units.

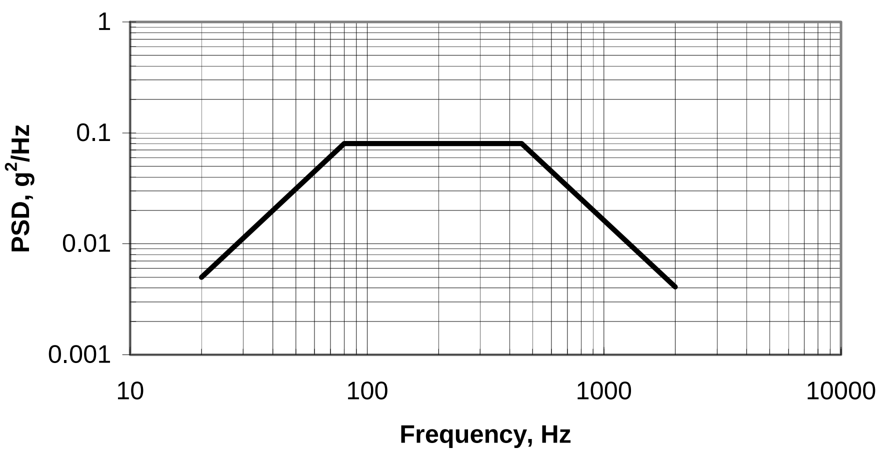

Figure 4 Assembly Random Vibration Test Acceleration profile

Table 1 Assembly Random Vibration Test Acceleration Inputs

Frequency (Hz)	Flight Acceptance Level	Qualification Level
20 – 80	+ 6 dB/octave	+ 6 dB/octave
80 – 450	0.04 g^2/Hz	0.08 g^2/Hz
450 – 2000	- 6 dB/octave	- 6 dB/octave
overall	5.5 g_{rms}	7.7 g_{rms}

5.2.4 Entry, Descent and Landing Induced Vibration

The TDS shall be designed to survive and operate under the random vibration environments specified in Table 2 simulating the aerodynamic loads during the Mars Entry and Descent and the pressure loading due to the descent stage thrusters. The TDS shall be powered on during the test for 2 minutes per axis for the qualification unit and 1 minute per axis for the flight units

Table 2 Entry and Descent Assembly Random Vibration Test Acceleration Inputs

Frequency, Hz	Flight Acceptance Level	Qualification/ Protoflight Level
20 - 2000	0.0004 g^2/Hz	0.0008 g^2/Hz
Overall	0.89 g_{rms}	1.26 g_{rms}

5.2.5 Pyrotechnic Shock

The TDS shall be designed to withstand 3 mechanical shocks per axis with the TDS turned on. The TDS shall be able to operate during the pyrotechnic shock event (may not have to meet performance specification) and be able to recover after the event without problems.

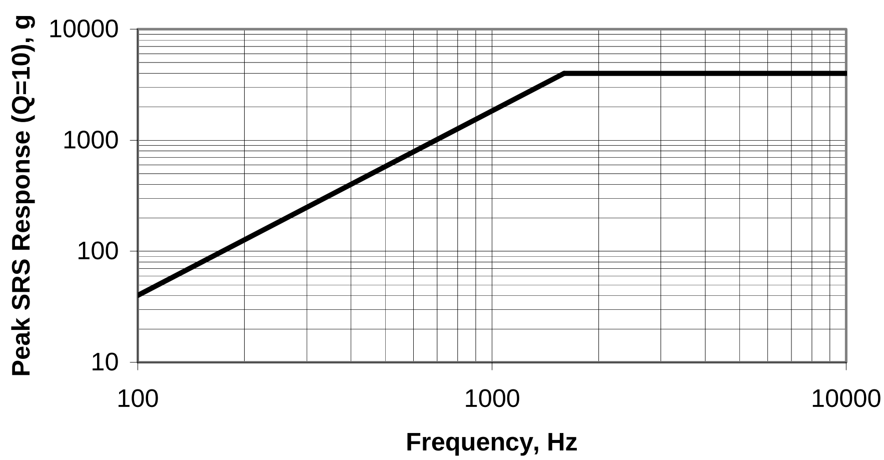

Figure 5 Pyrotechnic Shock Test Acceleration profile

Table 3 Pyrotechnic Shock Test Acceleration Inputs

Frequency (Hz)	QUAL, PF Peak SRS Response (Q = 10)
100	40 g
100 – 1600	+ 10 dB/octave
1600 – 10000	4000 g

5.2.6 High Energy Radiation Environment

The TDS shall not be damaged or reduced in performance in any way after exposure to the following high energy radiation environment:

5.2.6.1 Fluence

Up to 8.5×10^{10} neutrons/sec^2

5.2.6.2 Single Event Effects

Refer to Section 4.9 of JPL D-27175 (MAP)

5.2.6.3 Total Ionizing Dose

Up to 4.6 kRad

5.3 FLIGHT CRUISE ENVIRONMENTS

The TDS shall not be degraded due to 1 year non-operating storage in a space environment (1 x 10^{-14} torr) or a Mars environment (2-12 torr carbon dioxide) at temperatures of -100°C to +50°C. The spacecraft may provide heater, if necessary, to keep the TDS to a minimum temperature of -40ºC.

5.4 GROUND ENVIRONMENTS

5.4.1 Pre-Launch Storage

The TDS shall not be degraded due to long term storage in a clean, 75% relative humidity maximum, controlled environment. The unit shall be designed with a 5-year minimum shelf life after delivery under the above conditions. No storage life limiting parts, materials, or processes shall be used in the unit.

5.4.2 Contamination Control Bakeout

The TDS shall not be degraded due to exposure to the Contamination Control Bakeout environment called-out in Section 6.5.5 of JPL D-27175 (MAP). This bakeout is performed with the TDS powered off.

5.4.3 Planetary Protection Bakeout

The TDS shall not be degraded due to exposure to the Planetary Protection Bakeout at 110ºC for 125 hours or 125ºC for 12.5 hours. Contamination Control Bakeout may be combined with Planetary Protection Bakeout. This bakeout is performed with the TDS powered off.

5.4.4 Transportation and Handling

Provisions for packaging and other safeguards shall be such that the unit will not experience environmental conditions more severe than specified in Section 4.3 of JPL D-21382 (ERD).

6 MISSION ASSURANCE REQUIREMENTS

6.1 GENERAL

The TDS shall meet all the requirements of JPL D-27175 (MAP) (including EEE Parts, Hardware Quality Assurance, Software Quality Assurance, Reliability Assurance, Problem/Failure Reporting, Materials and Processes Control, and Contamination Control) with the following exceptions:

6.2 PARTS AND RELIABILITY

6.2.1 EEE Parts

Standard parts shall be defined as those that meet or exceed the following reliability standards shown in Table 4.

Table 4. Standard Parts

	Parts Reliability Standard	Minimum "Upgrade" Requirement
1)	MIL-PRF-38534 Class K level, QML Source	Post lead-form Hermeticity (if applicable)
2)	MIL-PRF-38535 Class V level, QML Source	Post lead-form Hermeticity (if applicable)
3)	MIL-PRF-19500 JANS level, QPL Source	Post lead-form Hermeticity (if applicable) Operating Life test on flight lot, if not already required per applicable military specification (45,000 device hour with a minimum of 22 samples)
4)	Military Established Reliability (ER) passive devices, Failure	DPA on crystals, filter, ceramic capacitors (except MIL-PRF-123), mechanical relays,

	Rate Level S or R or better such as Weibull B	and inductors. Surge current test on solid tantalum capacitors Ceramic capacitors rated at less than 100 V and used in less than 10 V applications shall be subjected to DPA. Dielectric thickness shall be verified to a minimum of 0.8 mils. Post lead-form Hermeticity (if applicable)

6.2.2 Non-Standard Parts

EEE parts not meeting the minimum quality and reliability criteria of standard parts in 6.2.1 are defined as non-standard parts. Unique, custom parts (e.g. Application Specific Integrated Circuits (ASICs), custom hybrids) and commercial parts are considered non-standard. Non-Standard Part Approval Request (NSPAR), or JPL approved equivalent, must be submitted to JPL for approval for all nonstandard parts. All non-standard parts shall be upgraded/screened to the requirements of Table 5, unless otherwise specified on a NSPAR as approved by JPL Parts Engineering and Radiation Specialists.

Plastic encapsulated microcircuits (PEMs) used in flight hardware shall be screened and qualified using JPL D-19426, Plastic Encapsulated Microcircuits (PEMs) Reliability/Usage Guidelines for Space Applications. All sample testing shall have sample sizes determined by the requirements in SSQ 25001 for a confidence level of at least 90% or as allowed by the Parts Specialist, with the concurrence of the JPL PIE.

Table 5 Upscreening for Non-Standard Parts

	Parts Reliability Standard	Minimum “Upgrade” Requirements [1/]
1)	MIL-PRF-38535, Class Q level, QML Source	P.I.N.D. Radiographic inspection RGA DPA Post lead-form Hermeticity (if applicable) Operating Life test on flight lot, if not already required per applicable military specification
2)	MIL-PRF-19500, JANTXV level, QPL Source	P.I.N.D. on cavity devices Radiographic inspection RGA on cavity devices

		DPA Post lead-form Hermeticity (if applicable) Operating Life test on flight lot, if not already required per applicable military specification
4)	All other nonstandard parts	100% screening and lot sample testing, including life test, as listed in this table for the nearest equivalent general military specification.

1/ Contractor will include the upscreening and lot sample testing listed in this table on the NSPAR submitted for JPL approval. Deviations from this table will be allowed if approved by JPL PIE via the NSPAR.

Nonstandard semiconductor parts shall require sample operational life test on the flight lot. For all parts life test shall be 22 pieces for 2000 hours at the device burn-in temperature, or the equivalent number of device hours calculated for modified sample size, test duration, and/or test temperature. Reduction in the equivalent life test device-hour requirement (for example, on high cost device types) shall be allowed if approved by the JPL PPM, but shall include a minimum of 5 pieces.

6.2.3 Reliability Assurance Plan

The TDS subcontractor shall submit a Reliability Assurance Plan in accordance with the Reliability Assurance conditions and requirements contained in JPL D-27175 (MAP) as required by Exhibit I and Exhibit II.

6.2.4 Reliability Analyses

Data and analyses to be submitted prior to PDR for review and again (as an updated final version) prior to CDR for JPL approval, as defined by the approved Reliability Assurance Plan in accordance with JPL D-5703, or other JPL- approved source and documented as required by Exhibit I and Exhibit II:

a. Electronic parts stress analyses at qualification test temperature (derating parameters from JPL D-8545 or other approved source)
b. Thermal and structural stress analyses
c. Worst case circuit analyses at qualification test temperature
d. Worst case power supply analysis
e. Single event effects (SEE) analyses, including single event upset, single event latchup, single event transient
f. GSE-to-TDS and TDS-to-spacecraft FMECAs

6.2.5 Problem/Failure Reporting Plan

The Contractor shall submit a Problem/Failure Reporting Plan in accordance with the P/FR conditions and requirements contained in JPL D-27175 (MAP) as required by Exhibit I and Exhibit II.

6.2.6 Quality Assurance

The TDS shall undergo surveillance during fabrication, assembly, and testing. All inspections and testing may be subject to witness by a JPL Quality Assurance representative. Specific inspections and tests shall require inspection as follows.

6.2.7 Manufacturing, Inspection, and Test Flow Plan

A Manufacturing, Inspection, and Test Flow Plan shall be prepared to show all inspection and test points.

6.3 TESTS

The following inspections and tests shall be performed on all delivered TDS hardware:

6.3.1 Post-Fabrication Functional

This test shall be performed to determine that all of the individual functions of the TDS operate properly. Each section of the circuit will be excited with appropriate waveforms and the resulting waveforms or data transfers will be recorded.

6.3.2 Regression Test Functional

This test is required in the event anomalies were found in the testing of paragraph 6.3.1 and rework of the assembly was necessary. Similar testing shall be performed as in paragraph 6.3.1, but with reduced combinations. The purpose of this test is to verify the success of the rework and to determine that no damage was done to other portions of the assembly during the rework process.

6.4 QUALIFICATION TESTING

Qualification testing shall include contamination and planetary protection bakeout (paragraph 6.8), thermal cycle life test, pyrotechnic shock test, and all tests included in the Flight Acceptance Tests.

6.4.1 Thermal Cycle Life Test

A thermal cycle life test of 24 cycles over the temperature range specified in 5.2.1.3 shall be performed on the qualification unit with no damage or significant deterioration to the performance and mechanical integrity (e.g. solder joints) of the unit. This test may be done in ambient pressure.

6.4.2 Pyrotechnic Shock

The TDS Qualification Unit shall be subject to a total of 9 shocks with the unit powered on to verify the pyrotechnic shock specification in paragraph 5.2.5.

6.5 FLIGHT ACCEPTANCE TESTING

Each TDS Qualification Unit and Flight Unit, upon completion of assembly, shall be subjected to acceptance testing, with JPL Quality Assurance surveillance. Acceptance testing shall consist of the following inspections and tests, performed in the sequence shown:

	Test	Test Method Paragraph
a.	Initial Inspection	6.7.1
b.	Complete Functional Test	6.7.2
c.	Burn-In Test	6.7.3
d.	Random Vibration Test	6.7.4
e.	Final Functional Test	6.7.5
f.	Final Examination	6.7.6

6.5.1 Flight Acceptance Test Procedure

A detailed Acceptance Test Procedure using the contractor supplied automated test equipment shall be prepared and approved by the JPL cognizant engineer and the JPL quality assurance representative prior to the start of testing. The procedure shall contain detailed testing instructions including specific electrical connections along with diagrams. Data sheets for recording the test results will be provided by the automated test equipment. Hard copies as well as electronic files of the completed data sheets shall be submitted with each unit as part of the end item data package.

6.5.2 Acceptance Test Failures

Acceptance tests shall be accomplished without any repairs or adjustments being performed on the TDS. If the TDS fails to meet the requirements of this specification, the TDS shall not be accepted. Testing shall be stopped and the JPL cognizant engineer and the JPL cognizant quality assurance representative shall be notified. Problem/failure reporting shall be consistent with the approved Problem/Failure Reporting Plan stated in paragraph 3.4.3 of JPL D-21382 (ERD).

6.6 TEST FACILITIES AND CONDITIONS

6.6.1 Test Facilities

The facilities used for testing the TDS shall provide the environments specified herein with sufficient margin to enable completion of specific tests without requiring removal or relocation of the test specimen during any one test. All testing for the TDS shall be performed in enclosed areas that provide protection from contamination and environments more severe than those specified herein.

6.6.2 Test Equipment
The TDS test procedures shall identify all test equipment that is to be used in the performance of TDS testing.

6.6.3 Instrumentation Accuracy
The test equipment used in performance of the tests specified herein shall be calibrated and certified for accuracy at least one order of magnitude greater than the tolerance limits specified.

6.6.4 Atmospheric Conditions
Unless otherwise specified, all tests required by this specification shall be made at an atmospheric pressure of 710 to 760 Torr at a temperature of 22 ± 5°C with a relative humidity of 30 to 70 percent.

6.6.5 Special Test Fixtures
All TDS test fixture designs shall be documented and subject to design change control. All test fixtures shall be identified in the applicable test procedure.

6.6.6 Test Records
All test data shall be identified with applicable unit serial number, test procedure paragraph, environmental condition, date and time of test, test software revision and filename, relative humidity, unit on time, and any performance measurements.

6.7 FLIGHT ACCEPTANCE TEST METHODS

6.7.1 Inspections (Pre-Acceptance Test)
The TDS and its applicable manufacturing records shall be examined to verify that the unit is in compliance with the following requirements:

Item	Requirement
Assembly records	applicable contract documents
Identification	4.7.2
Workmanship	4.7.7
Mass	4.5.2

6.7.2 Complete Functional
This test shall be performed to verify functional and performance specification using the approved acceptance test procedure. Any anomalous behavior will be recorded and isolated to the nearest interface point.

6.7.3 Burn-in Test
This test shall be performed at a vacuum level of 10^{-5} Torr or lower. The burn-in test shall test the TDS at maximum part stress levels at high temperature. The burn-in test temperature profile

is shown in Figure 6. The temperature limits and test duration for the Qualification and Flight Units are shown in Table 2.2 of JPL D-21382 (ERD). Contamination and Planetary Protection Bakeout requirements (with the TDS powered off) may be combined with the Burn-in test. All test functions will be performed and monitored by the test equipment. During the power-on thermal cycling phase of the test, TDS measurements shall be taken at 10ºC intervals to detect any temperature-dependent anomaly. Any anomalous behavior shall be recorded and isolated to the assembly serial number and the nearest interface point.

6.7.4 Random Vibration Test

The TDS shall be subjected to a random vibration test per the requirements of paragraphs 5.2.3 and 5.2.4. This testing is performed following the high temperature burn-in test specified in paragraph 6.7.3. The random vibration level and duration for the test with the TDS powered off are specified in paragraph 5.2.3 and with the TDS powered on are specified in paragraph 5.2.4. Input accelerometers shall be mounted on all three axes of the holding fixture and monitored on a suitable recorder during all vibration tests.

6.7.5 Final Functional Test

This test is performed following the random vibration testing specified in paragraph 6.7.4. This test shall be performed using the same test equipment for the functional test (paragraph 6.7.2) with the unit under test installed in a thermal chamber. A vacuum chamber is not required, but may be desired to prevent frosting and significant heat loss to the ambient environment. The testing procedure will verify that the TDS meets all functional and performance specification. Any anomalous behavior will be recorded and isolated to the nearest interface point. An anomaly record will be produced and disposition will be determined on an individual basis.

6.7.6 Final Examination

The TDS, including all test records, shall be inspected at the completion of all acceptance testing through paragraph 6.7.5. The TDS and its documentation shall be in accordance with the requirements of this specification.

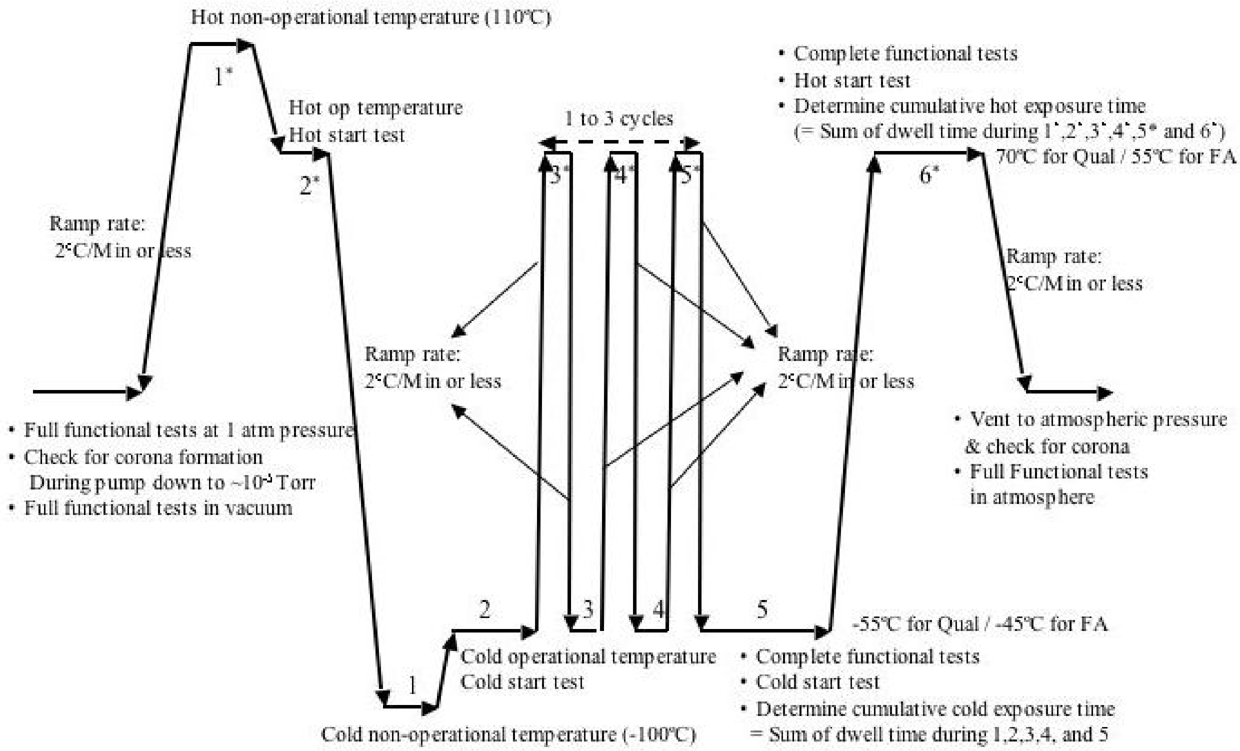

Figure 6 TDS Flight Acceptance Thermal Test Profile

6.8 CONTAMINATION AND PLANETARY PROTECTION BAKEOUT

This bakeout is performed on the Qualification unit and the Flight Units to meet the requirements of Contamination Control in paragraph 5.4.2 and Planetary Protection in paragraph 5.4.3. This bakeout may be combined with the Burn-in Test in paragraph 6.7.3. The data recorded during the entire bakeout shall include the vacuum level of the test chamber, the temperature of the test hardware, and the elapsed test time. The hardware shall be protected from recontamination from this point until delivery to the user.

6.9 PLANETARY PROTECTION

The TDS shall be subjected to a dry heat microbial reduction decontamination environment for the purposes of protecting Mars from contamination by Earth bound organisms. The Planetary Protection Bakeout shall be exposure to a vacuum environment (1 x 10^{-5} torr or better) at a temperature of 110ºC for 125 hours or 125ºC for 12.5 hours. Once the hardware has been exposed to this environment for the purposes of planetary protection, extreme care shall be exercised to prevent recontamination.

Appendix A Abbreviations (applies to Exhibit III and the Statement of Work)

ASIC	Application Specific Integrated Circuits
ATLO	Assembly, Test, and Launch Operations
CDR	Critical Design Review
CDRL	Contract Data Requirements List
CM	Configuration Management
COTS	Commercial, Off-The-Shelf
CTM	Contract Technical Manager
D	JPL Document (D-xxxx)
dB	Decibel
DPA	Destructive Physical Analysis
DRD	Data Requirement Description
E	Entry (time)
EDL	Entry, Descent, Landing
EEE	Electrical, Electronic, and Electromechanical
EM	Engineering Model
EMC	Electromagnetic Compliance
ER	Established Reliability
ERD	Environmental Requirements Document
FA	Flight Acceptance
FMEA	Failure Mode and Effect Analysis
FMECA	Failure Mode Effects and Criticality Analysis
GSE	Ground Support Equipment
ICD	Interface Control Document
IMU	Inertial Measurement Unit
JAN	Joint Army/Navy
JANS	JAN—most Stringent reliability level
JANTX	Joint Army/Navy Extra Testing requirements
JANTXV	JANTX with Visual inspection requirements
JSC	Johnson Space Center
JPL	Jet Propulsion Laboratory
L/D	Lift-to-Drag ratio
MAP	Mission Assurance Plan
MMR	Monthly Management Review
MRB	Material Review Board
MSL	Mars Science Laboratory
NASA	National Aeronautics and Space Administration
OSS	(NASA) Office of Space Science
NSPAR	Non-standard Part Approval Request
PDR	Preliminary Design Review
PEM	Plastic Encapsulated Microcircuits

PF	ProtoFlight
P/FR	Problem/Failure Report
PIE	Parts Interface Engineering
P.I.N.D.	Particle Impact Noise Detection
PP	Planetary Protection
PPM	Project Procurement Manager
PSD	Power Spectral Density
Qual.	Qualification
QML	Qualified Manufacturers List
QPL	Qualified Products List
RGA	Residual Gas Analysis
rms	root-mean-square
S/C	Spacecraft
SE	Support Equipment
SEE	Single Event Effects
SRS	Shock Response Spectrum
SSQ	DPA Testing for the Space Station
TBD	To Be Determined
TDS	Terminal Descent Sensor
TML	Total Mass Loss
URL	Uniform Resource Locator
VCM	Volatile Condensable Materials
VDC	Volts, Direct Current

Appendix B Preliminary Requirements Verification Table

Paragraph	Title	Similarity	Analysis	Inspection	FA Test	QUAL Test
4	Instrument Requirements					
4.1	Functional Requirements					
4.1.1	System Architecture			X		
4.1.2	Beam Configuration				X	
4.1.3	Beam Width		X		X	
4.1.4	Antenna Architecture			X		
4.1.5	Velocimetry		X		X	X
4.1.6	Altimetry		X		X	X
4.1.7	Output		X		X	X
4.1.8	Independent Operation				X	X
4.1.9	Interface				X	X
4.2	Performance Requirements					
4.2.1	Instantaneous Velocity Measurement Noise					
4.2.1.1	Vx		X		X	X
4.2.1.2	Vy		X		X	X
4.2.1.3	Vz		X		X	X
4.2.2	Bias Speed Offset		X		X	X
4.2.3	Bias Scale Factor		X		X	X
4.2.4	Quantization Accuracy					
4.2.4.1	Velocimetry Quantization				X	X
4.2.4.2	Altimetry Quantization				X	X
4.2.5	Instantaneous Slant Range Measurement Error		X		X	X
4.2.6	Beam Pointing Knowledge		X			
4.2.7	Acquisition Time		X		X	X
4.2.8	Output Rate				X	X

4.2.9	Time Tag Knowledge Error	X	X	X
4.2.10	Measurement Latency	X	X	X
4.3	Operational Envelope			
4.3.1	Altitude Range	X	X	X
4.3.2	Vertical Velocity Range	X	X	X
4.3.3	Horizontal Velocity Range	X	X	X
4.3.4	Attitude			
4.3.4.1	For 3000 m to 2000 m altitude	X	X	
4.3.4.2	For 2000 m to 1000 m altitude	X	X	
4.3.4.3	For 1000 m to 10 m	X	X	
4.3.5	Attitude Rate Limits			
4.3.6	Linear Acceleration Limits			
4.3.6.1	Vertical Acceleration	X	X	X
4.3.6.2	Horizontal Acceleration	X		
4.3.7	Ground Slopes			
4.3.7.1	Long Wavelength	X		
4.3.7.2	Short Wavelength	X		
4.3.8	Unambiguous Altitude	X	X	X
4.3.9	Unambiguous Velocity	X		
4.4	Interface Requirements			
4.4.1	Velocimetry			
4.4.1.1	Time Tag		X	X
4.4.1.2	Validity Flag		X	X
4.4.1.3	Beam Number		X	X
4.4.1.4	Velocity Measurement		X	X
4.4.2	Altimetry		X	
4.4.2.1	Time Tag		X	X
4.4.2.2	Validity Flag		X	X
4.4.2.3	Beam Number		X	X
4.4.2.4	Slant Range Measurement		X	X

4.4.3	Data Interface Standard			X	X
4.4.4	Bus Voltage			X	X
4.5	Physical Requirements				
4.5.1	Dimensions				
4.5.1.1	Footprint		X		
4.5.1.2	Height		X		
4.5.2	Mass		X		
4.5.3	Power			X	X
4.6	Life Expectancy				
4.6.1	Operating Life	X			
4.6.2	Thermal Cycle Life	X		X	X
4.6.3	Shelf Life	X			
4.7	MECHANICAL REQUIREMENTS				
4.7.1	Alignment Markings		X		
4.7.2	Identification and Marking Methods		X		
4.7.3	Connectors		X		
4.7.4	Connector Retention		X		
4.7.5	Venting	X			
4.7.6	Maintainability		X		
4.7.7	Workmanship		X		
4.7.8	Contamination Control		X		
4.8	ELECTRICAL REQUIREMENTS				
4.8.1	Conducted Emissions				X
4.8.2	Radiated Emissions				X
4.8.3	Radiated Susceptibility				X
4.8.4	Radiated Susceptibility				X
4.8.5	Rejection of Undesired Signals				X
4.8.6	Power Line Ramp Voltage				X
4.8.7	Bonding, Isolation, and Grounding				X
4.8.8	Shielding		X		
4.8.9	Cabling		X		

4.9	SAFETY					
4.9.1	Design Principles					
4.9.2	System Standards					
5	ENVIRONMENTS					
5.1	MARS ENVIRONMENTS					
5.1.1	Operational Flight Temperature				X	X
5.1.2	Barometric Pressure				X	X
5.1.3	Relative Humidity				X	X
5.1.4	Atmospheric Composition		X			
5.1.5	Radar Backscatter Characteristics		X			
5.2	ENVIRONMENTAL DESIGN REQUIREMENTS					
5.2.1	Operational Temperature Limits				X	X
5.2.1.1	Allowable Flight Temperature				X	X
5.2.1.2	Flight Acceptance (FA)				X	X
5.2.1.3	Qualification (Qual)					X
5.2.2	Thermal Vacuum				X	X
5.2.3	Random Vibration				X	X
5.2.4	Entry, Descent and Landing Vibration				X	X
5.2.5	Pyrotechnic Shock				X	X
5.2.6	High Energy Radiation Environment	X	X			
5.2.6.1	Fluence	X	X			
5.2.6.2	Single Event Effects	X	X			
5.2.6.3	Total Ionizing Dose	X	X			
5.3	FLIGHT CRUISE ENVIRONMENTS					
5.4	GROUND ENVIRONMENTS					
5.4.1	Pre-Launch Storage			X		
5.4.2	Contamination Control Bakeout			X		
5.4.3	Planetary Protection			X		

	Bakeout					
5.4.4	Transportation and Handling			X		
6	MISSION ASSURANCE REQUIREMENTS					
6.1	GENERAL			X		
6.2	PARTS AND RELIABILITY					
6.2.1	EEE Parts		X	X		
6.2.2	Non-Standard Parts		X	X		
6.2.3	Reliability Assurance Plan			X		
6.2.4	Reliability Analyses		X			
6.2.5	Problem/Failure Reporting Plan			X		
6.2.6	Quality Assurance	X		X		
6.2.7	Manufacturing, Inspection and Test Flow Plan			X		
6.3	Test					
6.3.1	Post-Fabrication Functional				X	X
6.3.2	Regression Test Functional				X	X
6.4	QUALIFICATION TESTING					
6.4.1	Thermal Cycle Life Test				X	X
6.4.2	Pyrotechnic Shock					X
6.5	FLIGHT ACCEPTANCE TESTING					
6.5.1	Flight Acceptance Test Procedure			X		
6.5.2	Acceptance Test Failures			X		
6.6	TEST FACILITIES AND CONDITIONS					
6.6.1	Test Facilities			X		
6.6.2	Test Equipment			X		
6.6.3	Instrumentation Accuracy			X		

6.6.4	Atmospheric Conditions	X		
6.6.5	Special Test Fixtures	X		
6.6.6	Test Records	X		
6.7	FLIGHT ACCEPTANCE TEST METHODS			
6.7.1	Inspection (Pre-Acceptance Test)	X		
6.7.2	Complete Functional		X	X
6.7.3	Burn-in Test		X	X
6.7.4	Random Vibration Test		X	X
6.7.5	Final Functional Test		X	X
6.7.6	Final Examination	X		
6.8	Contamination and Planetary Protection Bakeout		X	
6.9	Planetary Protection		X	